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Comparison of Requirements for Composite Structures for Aircraft and Space Applications

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Abstract

In this paper, the aircraft and space vehicle requirements for composite structures are compared. It is a valuable exercise to study composite structural design approaches used in the airframe industry, and to adopt methodology that is applicable for space vehicles. The missions, environments, analysis methods, analysis validation approaches, testing programs, build quantities, inspection, and maintenance procedures used by the airframe industry, in general, are not transferable to spaceflight hardware. Therefore, while the application of composite design approaches from other industries is appealing, many aspects cannot be directly utilized. Nevertheless, experiences and research for composite aircraft structures may be of use in unexpected arenas as space exploration technology develops, and so continued technology exchanges are encouraged.

Introduction

Aircraft and space vehicle structures are performance structures. Because of their light weight and high strength, composites are potential candidates for these structures [1]. They are widely used in secondary structures and are currently making their way into aircraft primary structures such as wings and fuselage components (Boeing 787 Dreamliner and Airbus A380 aircrafts.) While space vehicles are behind in the use of composites, composites are considered attractive alternatives to metallic structures. When new materials are considered for structural applications, one of the first issues to confront would be the requirements for these structures and structural components [2]. The purpose of this paper is to attempt a comparison of the requirements for composite structures for aircraft and space applications.

The paper is organized as follows. First, a general discussion on special features of composites is presented. This is followed by a general comparison of features of aircraft and spacecraft missions. Then, the comparison of requirements for design allowables, development of these allowables, factors of safety, structural integrity, and damage tolerance is presented. For convenience in presentation, spacecraft and space vehicle structures are used interchangeably.

Composites – General Features

Compared to the widely used metallic materials, composites differ significantly in many characteristics. Some of these special features of composites are discussed here.

- Composites demonstrate high strength-to-weight ratios.
- Composites exhibit higher dimensional stability, corrosion resistance, and thermal cycling capability.
- Composites provide a numerous of options for structural tailoring for known loading conditions.
- Constituent materials in the composite strongly influence composite structure's stiffness, strength, and durability.
- Composites have practically no ductility. Most composites are brittle and have nearly linear stress-strain law to failure.
- Composites are non-isotropic.
- Composites have a strong link between manufacturing processes and the subsequent mechanical properties. Therefore one needs to pay special attention to the manufacturing processes, realize that there can be wide variability in the allowables data, and realize the need for a careful building-block test program to develop the design allowable data.
- Composites are highly sensitive to notches. Stress concentration factors higher than 10 are possible for some notches, and the stress gradients near the notches are very high.
- In composites, large in-plane stress gradients develop near geometric and material discontinuities. In-plane stress gradients lead to out-of-plane or interlaminar stresses. Because composites have low through-the-thickness strength, delaminations can initiate

and grow. Delaminations degrade the strength of the composite laminate and can lead to failure. Figure 1 shows some typical areas where delaminations initiate because of geometric and material discontinuities.

- Composites degrade due to moisture, humidity, and heat. The coefficient of thermal expansion along the fiber direction is near zero or a small negative number compared to the other two transverse directions.
- Composites are sensitive to accidental damages. There is a significant loss of strength due to low-velocity impacts. Figure 2 shows the dramatic reduction in compression after impact strength as a function of impact energy for a commonly used composite system. In thick composites, the damage due to these impacts is in the interior the laminate and is either not visible on the outer surfaces of the laminate or is barely visible.
- Composites have low-sensitivity to fatigue for in-plane loading.

Comparison of Mission Requirements

The commercial and military aircraft industries base their requirements on the performance required out of the appropriate certifying authority. Commercial transport aircraft usually have a design life of 75,000 cycles (take-offs and landings) over 30 years. Military transport aircraft may have similar operational life. Many copies, perhaps thousands of copies, of commercial aircraft are built, while tens to a few hundred copies of military aircraft are built. The manufacturing processes are highly specialized by suppliers through extensive testing of company designs of materials, coupons, joints, components, and structural elements. As a result of this comprehensive testing, an extensive database of material properties and structural performance data is created for each system. The aircrafts are certified to meet applicable requirements based on detailed reviews that ensure that the aircrafts meet safety and performance criteria for the fleet mission environments and loadings, which are usually well defined [3-5] and can be incorporated in the certification test programs. In addition, the aircraft applications rely on extensive development programs that continue into the operational and maintenance regimes of the mission lifetime. These programs ensure continued airworthiness through periodic inspections that may include depot re-work using certified repair and/or structural upgrades as needed throughout the mission lifetime (see Refs. 3-5).

In contrast, space vehicle missions typically have no or very limited opportunities to inspect, repair, or modify the structure once they lift off from the launch pad. This is a major and fundamental difference between the aircraft approach and the spacecraft approach to composite structural design methodology. While the aircraft in-service inspection/repair approach could be adopted for any space vehicle that returns periodically to Earth, it would require the space vehicle program to develop and certify capabilities similar to those in use for aircraft while addressing the more challenging aspects of missions that operate in space environments. There are some examples within the NASA Space Shuttle Program where refurbishment of vehicle components is performed, but this practice is not common because of the risks of damage to expensive space-qualified vehicles. Instead, spacecraft structures are typically designed for a single launch and certified to perform their entire mission without interim inspections or repairs to reduce operational costs.

Space vehicles experience unique environments and loadings, and there will be few opportunities to validate the vehicle's structural performance under the mission conditions. Therefore, the space vehicle structural design must be more robust than a design obtained using comparable aircraft structural design procedures. Furthermore, at most, only a few copies of the space vehicle structure will probably be built – perhaps more launch vehicles than crew modules and even fewer lunar and interplanetary systems. Thus while some aspects of the aircraft industry's composite design best practices (such as those for ply drops, stacking sequences, fiber/resin combinations, etc.) may benefit composite spacecraft structural design, most of their best practices may not be directly applicable to spacecraft composite structures. Nevertheless, a careful comparison of requirements for aircraft and spacecraft composite structures is very useful.

Comparison of Specific Requirements

Five major requirements for both aircraft and spacecraft structures are discussed and compared. The requirements are focused on the standard practices as they are followed today for composite structures.

I. Relevant Design Allowables:

Strength allowables are consistent between the aircraft and spacecraft applications. Both aircraft and spacecraft standards require that strength allowables should properly account for the effects of environment (e.g., temperature, humidity, chemicals), but beyond that they do not specify which properties (e.g., notched or unnotched strength) should be used for design allowables.

Strength allowable values are generated typically using uniaxial tests performed at different environmental conditions (e.g., different values of temperature, humidity). These data are usually from load-to-failure tests with maximum load ('stress') and strain-at-failure recorded. For composite systems exhibiting a nonlinear response, repeated loading-unloading cycles with increasing load level may provide insight into the overall material response (i.e. nonlinear elastic or damage accumulation).

The definition of statistical sampling and data reduction methods for what constitutes an A- or B-basis for strength allowables is common among the aircraft and spacecraft design standards. Also common is the general requirement that B-basis design allowables will be used for redundant or fail-safe structure and A-Basis design allowables will be used for single load path structure. A redundant or fail safe structure is defined as follows: A structural design criterion in which it must be shown that the structure remaining, after failure of any single structural member, can withstand the resulting redistributed internal limit loads without failure. The ability to sustain a failure and retain the capability to safely terminate or control the operation. Thus redundant fail-safe structural designs are desirable. In fact, space vehicle fail-safe structural designs using B-basis strength allowables are used by the current NASA [6-9] and European Space Agency (ESA) requirements [10-12].

Joint designs, whether bolted or bonded or both, are complex structural components. Load paths and reactions are dependent on many factors associated with the joint itself. Adding redundancy in the joint system does add complexity with respect to load path and other geometric limitation along with additional mass. Incorporating redundancy in a specific joint design to give fail-safe capability and migrate the A-basis allowable requirement to a B-basis requirement may be as difficult or costly to achieve as establishing an A-basis allowable for the specific joint configuration. The interaction of other joint factors or fitting factors also needs to be reviewed in the designs. The airframe industry typically establishes statistically relevant performance data for each major joint configuration through comprehensive test programs. The industry also maintains heritage configurations and databases for each design. While it may be desired, it is unlikely that such an approach will be used for spacecraft structures.

Probabilistic or non-deterministic design approaches are being proposed and pose an additional paradox. Without statistically relevant data, mean values, standard deviations, and distribution forms for design allowables will have to be assumed and this would result in uncertainties in the analytical predictions. These predictions can be used to determine the relative sensitivity of selected response variables within assumed ranges and distributions. This paradox will most likely lead to specific focused building-block test programs for key design-driving parameters.

II. Development of Relevant Design Allowables:

A building-block approach starts by studying simple basic configurations and then builds in complexities associated with the design in a systematic and progressive manner. A basic building-block program can have different objectives, goals, and scope depending on the application and the program. In the most cases, the objective of the building-block approach is to establish statistically relevant design allowables through an extensive test program moving from coupon-level specimens to component-level tests and even to full-scale tests. In other cases, the objective of the building-block approach is to define the composite structural strength and to identify and characterize failure modes using a limited number of specimens. In yet other cases, the building-block information must be sufficient to guide the definition of a meaningful full-scale/component test – including consideration for appropriate type/size/amount of damage [1,13]. Figure 3 and 4 demonstrate how this is practiced currently for aircraft structures. For composite structures design for aircraft and spacecraft, multiple types of building-block programs may be required to develop robust composite designs that fully exploit the potential of each composite material system.

While the building-block approach has all the aforementioned advantages, the approach has some disadvantages. A “properly executed” building-block approach as used by the airframe industry may not be practical for space vehicle designs from a cost-schedule perspective. The airframe industry’s building-block approach usually involves testing of hundreds of specimens. For the airframe industry, significant R&D resources are put into understanding their specific structural systems and generating the necessary databases sufficient for design and certification. The resource cost of this effort can be amortized across the fleet of vehicles produced.

Unfortunately, the space vehicle industry only produces a limited number of products making the institution of an extensive building-block test program impractical from a cost perspective.

Often spacecraft structural systems are unique or application dependent. Design allowable data from a building-block approach may not be available early enough in the design cycle to affect the robustness or efficiency of the spacecraft design. A generalized building-block test program may produce design data and expose structural weaknesses. However, unless the test program performs testing of *all* material, configuration, and loading complexities, some structural weaknesses may not be recognized before a flight mission or before a catastrophic failure at worst, or before large-scale damage testing results are obtained. Furthermore, this building-block approach could potentially lead to major program redesign/development cost increases and schedule delays. Finally, due to the extreme complexity or impossibility of *a priori* knowledge and simulation capability of the spacecraft operational environment on Earth, the development of a fully encompassing building-block program may not be achievable.

Clearly, traditional building-block approaches used by the aircraft industry are not directly applicable for space vehicles. For successful composite structures and structural components, an appropriate building-block approach needs to be designed and executed to exploit all the benefits, avoid the pit-falls, and mitigate known risks.

III. Factors of Safety:

The historic factor of safety (FOS) for aircraft structures is 1.5 [14], while for space vehicle structures it is 1.4 [15]. These FOS are used generally for most of the acreage areas of the structures exhibiting near uniform stress states, and specific parts have higher factors. There is a significant difference in the NASA approach to areas of discontinuities and this will be discussed below.

In the NASA-STD-5001A [15], “discontinuities are defined as an interruption in the physical structure or configuration of the part. Delaminations, debonds, and dropping of plies are all assumed to be discontinuities.” In addition, the term “discontinuity area” is defined in a recent structural design verification requirements document [6] as: “A local region of a composite or non-metallic structure consisting of built-up plies, chopped fiber or reinforced regions around fittings, joints or interfaces where the stress state and load distribution within the region may be difficult to characterize. A region is considered a discontinuity area until uniform section properties in the structure can be considered in the structural analysis. Bonded joints are considered discontinuities.”

In the preliminary design phase of aircraft or spacecraft, predictive analysis tools and techniques for local discontinuities in composite structures are difficult to incorporate (and validate) in the design process. At the present time, these tools and modeling practices are not sufficiently robust to provide adequate fidelity of the stress state in areas of discontinuities. Through-the-thickness or out-of-plane stresses may not be correctly characterized and understood, leading to incorrect characterization of these discontinuities. In part, the use of a discontinuity area FOS is applied to address the modeling and analysis uncertainties early in the design.

The discontinuity area FOS approach is suggested and defined for strength requirements in NASA-STD-5001A [15] and in the FAA Commercial Space Transportation guidance document for reusable launch and reentry vehicles [16]. FOS listed in NASA-STD-5001A [15] are stated as “minimum” values, and some structural members or systems may be required to meet other more stringent and restrictive performance requirements. Specific material systems and design configurations need to be tested to determine statistically relevant allowable data and characterization of failure modes. Such testing typically is performed as an integral part of a larger building-block test program for the overall design and is representative of current best practices.

A possible misconception within the technical community regarding FOS also needs to be addressed here. *The misconception is that the FOS covers uncertainty in loads and/or material properties.* The uncertainty in the loads is *not* covered by the FOS and needs to be accounted for by the use of appropriate load uncertainty factors (LUF). These LUFs depend on the environment in which the structure operates. In a similar manner, the uncertainty in material properties is *not* covered by the FOS. The uncertainties in the material properties need to be accounted for by the use of A-basis and B-basis properties for the operating environments. These FOS account for uncertainties and variations that cannot be analyzed or accounted for in a rational manner, including modeling uncertainties, analysis assumptions, defects in material and fabrication issues, and unknown/unexpected aspects of complex structures.

In summary, a comprehensive building-block approach to developing statistically relevant data is the correct technical approach for space vehicle composite structures. However, the practical application of such an approach may not be cost effective or feasible. On the other hand, when the building-block approach for composite spacecraft structures is not “properly executed” due to cost and schedule constraints, there is a risk of structural failures. A methodology that establishes a balance between the deterministic FOS approach and a sufficient building-block program is needed.

IV. Structural Integrity:

Generally, structural integrity is evaluated based on the following three factors:

- The maximum estimated structural response. The estimated or anticipated structural response is usually determined through established analysis and/or test methods.
- The uncertainty of the estimate. The uncertainty in the estimated structural response is established through best practice. NASA-STD-5001A [15] represents NASA’s view of the current “best practice” for establishing FOS for spaceflight hardware [17] and damage tolerance for composite structures is established using MSFC-RQMT-3479 [9]. However, these FOS values may be tailored by Programs provided an adequate engineering risk assessment is performed and approval is obtained by appropriate Technical Authority.
- The capability of the structure. The structural capability is generally established by material capabilities.

This paradigm is well accepted in the metallic structures community where analysis methods and material capabilities are well understood. However, in composite structural systems, the material capability is heavily influenced by the geometry, laminate stacking sequences, material architecture (uni-directional tape, fabric, textiles), manufacturing processes, and bulk material properties. These influencing factors lead to additional uncertainty in the structural integrity of composite systems. This uncertainty primarily pertains to structural regions, such as joints, variable thickness regions, stiffeners, or cut-outs, where the stress state or load path is difficult to assess using current analytical tools and methodologies and where the stress state is strongly dependent on the material definition, material properties, and fabrication. The airframe and space vehicle industries handle this additional uncertainty in different ways. The airframe industry maintains a consistent single FOS that covers general uncertainty (1.5) combined with a reliance on an extensive building-block test program to determine the structural capability. Such a program could involve hundreds of tests for a structural application using a specific material, fabrication processes, configurations, loadings, and damage in order to establish a statistical basis [18]. In contrast, the current NASA-STD-5001A [15] practice, as well as the FAA guidance document for reusable launch and reentry vehicles [16], relies on a dual factor of safety approach: one FOS is applicable to areas of the structure where the response is well understood (for example, 1.4 for uniform areas); and another FOS for areas of the structure that are not well understood (for example, 2.0 for discontinuity areas). However, the material properties are left to the individual programs to establish. NASA-STD-5001A states:

“designs are assumed to use materials having mechanical properties that are well characterized for the intended service environments and all design conditions ... Material allowables shall be chosen to minimize the probability of structural failure due to material variability ... Allowables shall be based on sufficient material tests to establish values on a statistical basis”

Again, these requirements are well understood and applied for metallic structural systems. However, they become problematic for composite systems because of the influencing factors previously mentioned. Hence, the airframe industry approach to developing material allowable data using a building-block approach appears attractive, but for spacecraft the development of statistically relevant data is not always feasible. To further complicate the issue, in using the deterministic approach, there is evidence [4,19] that a factor of 2.0 may not be conservative depending on the severity of manufacturing flaws, design features such as cutouts and holes, and/or impact damage. Higher FOS values may be warranted in some cases depending on the adequacy of the test database, the conservatism of the design allowable, inclusion of knockdown factors, and the method of design verification.

In situations where structurally similar design and fabrication processes have demonstrated reliable strength in well-defined loading/environment conditions, Section 4.4 of NASA-STD-5001A [15] has a provision to request an alternative approach by waiver when a substantial case can be made. This waiver would need to be approved by the responsible program authorities and technical authorities. Finally, in the event a structural weakness is discovered late in the structural development program or launch planning and provided testing establishes that the weakness is an acceptable risk, then this waiver process can be pursued. Note that such an approach is feasible only when the component was originally designed with the specified FOS.

V. Damage Tolerance:

For aircraft structures damage tolerance is achieved through a methodology developed by the FAA [4] and is shown in Figure 5. In this figure the design load level is plotted against increasing damage severity. On the abscissa, the allowable damage limit (ADL) and critical damage threshold (CDT) are plotted. The ADL corresponds to the limit of the Category 1 damage – “Barely Visible Impact Damage” (BVID) and “Allowed Manufacturing Damage”. The CDT corresponds to the damage limit when the residual strength of the structure falls below the design limit load (DLL). Note that aircraft composite structures are required to carry the design ultimate load (DUL) with any non-detectable damage. For reduced load capability with “Category 2 or 3” damage conditions, the structure is not permitted to have reduced strength below DLL. In addition, the flight mission is adjusted such that opportunities for certified inspection that detects the damage and certified repairs that restores the structure to ultimate load capability are provided. (In fact in determining the CDT, one missed inspection cycle is built in.) Category 4 damage, such as liberation of rotor blade and subsequent impact with the aircraft structure, bird strikes, lightning damage, hail damage etc., would cause a loss of limit load capability. However, in these situations, the pilot is aware of the damage and will execute a safe landing by adjusting the flight envelope to reduce flight loads.

The Category 1 damage indicated in Figure 5 that corresponds to space applications would likely include damage due to fabrication, assembly, ground handling, transportation, and any other sources of damage occurring prior to launch. According to the aircraft criteria, this damage must not decrease strength below ultimate strength, which is the same as the existing space vehicle requirement [9]. In addition, space missions must consider damage due to all other sources that may occur during a spacecraft mission that includes: liftoff, ascent, time spent in orbit from risk of impact by micrometeoroids or other space debris, the return to Earth, and subsequent missions. Additional unique damage risks may exist during long-duration lunar and interplanetary missions. These aspects are addressed in the damage threat assessments and the strength with damage requirements of Ref. 9. Damage from impact threats after launch and their consequences for space missions are uniquely different from those of the airframe industry.

No space mission conditions are known at this time that permit the use of the aircraft Category 2, 3, or 4 damage indicated in Figure 5 because inspection and repair capability are not feasible once the space vehicle is launched.

The NASA fracture control damage tolerance requirements for composites specified in Section 5.3.2.6 of Ref. 9 do not permit damage growth. This requirement was instituted to control risk of fracture in composites because technology for predicting damage initiation and accumulation for composites is not as reliable as it is for metals. This requirement is similar to that is currently practiced by the aircraft industry. A new fracture control standard, NASA-STD-5019A [20] is developed recently that is applicable to both metallic and composite materials. It is also expected that this standards development activity will offer opportunity for discussion of different approaches to composite spacecraft structural design requirements.

In summary, the key points are:

- Aircraft requirements are similar to space vehicle requirements to sustain ultimate loads with manufacturing and other defects/damage before launch.
- Aircraft requirements permit some strength reduction below ultimate for category 2 and 3 damage, but this is different from usual space applications because:
 - Aircraft practice specifies periodic depot inspections and repairs are performed to ensure continued airworthiness, and
 - Aircraft practice requires inspections and repairs to restore the structure to ultimate load capacity within a short fraction of the mission life between the required depot work.
- Space vehicle programs usually do not include development of certified inspection and repairs of structure to restore it to ultimate strength capacity after launch. Also, usually there are no opportunities for these activities.

Concluding Remarks

Composites are being widely used in aircraft structures because of their high strength-to-weight ratios. Their use in space applications is increasing. The requirements for composite structures for aircraft and space applications are compared and discussed. The requirements for design allowables, strength allowables, factors of safety, structural integrity, and damage tolerance are compared. To develop design allowables, the aircraft industry employs a building-block approach involving testing of hundreds of specimens. As space vehicles are only produced in extremely limited numbers, traditional building-block approaches such as that used by the aircraft industry are impractical. Requirements for strength allowables, on the other hand, are consistent between the aircraft and spacecraft applications. The requirements on factors of safety used by the spacecraft structures are more stringent than those for the aircraft structures. To ensure structural integrity, a comprehensive building-block approach to developing statistically relevant data is the approach used by both aircraft and spacecraft composite structures. For space applications a methodology that establishes a balance between factor of safety and building-block testing that is tailored for specific applications is needed. To ensure damage tolerance, aircraft and spacecraft follow similar requirements. Aircraft requirements are similar to space vehicle requirements to sustain ultimate loads with manufacturing and other defects/damage before launch. Aircraft practice requires periodic inspections and restoration of the structure to design ultimate load capacity many times during its mission life. Most space vehicles do not have such opportunities after launch and lift-off. Thus the damage tolerance requirements for aircraft and space vehicles are much different with only similarities up to the allowable damage limit.

Because of mission and operations differences, the requirements for composite structures for aircraft applications are not directly applicable as requirements for space vehicle applications. Space vehicles experience unique environments and loadings and there will be fewer, if any, opportunities to inspect and repair the vehicle's structure and validate its structural performance under mission conditions. Therefore, the space vehicle composite structural design must be more robust than the aircraft composite structural design. Nevertheless, it is a worthwhile endeavor to examine the aircraft industry experience and knowledge base for possible adaptability to spacecraft composite structures.

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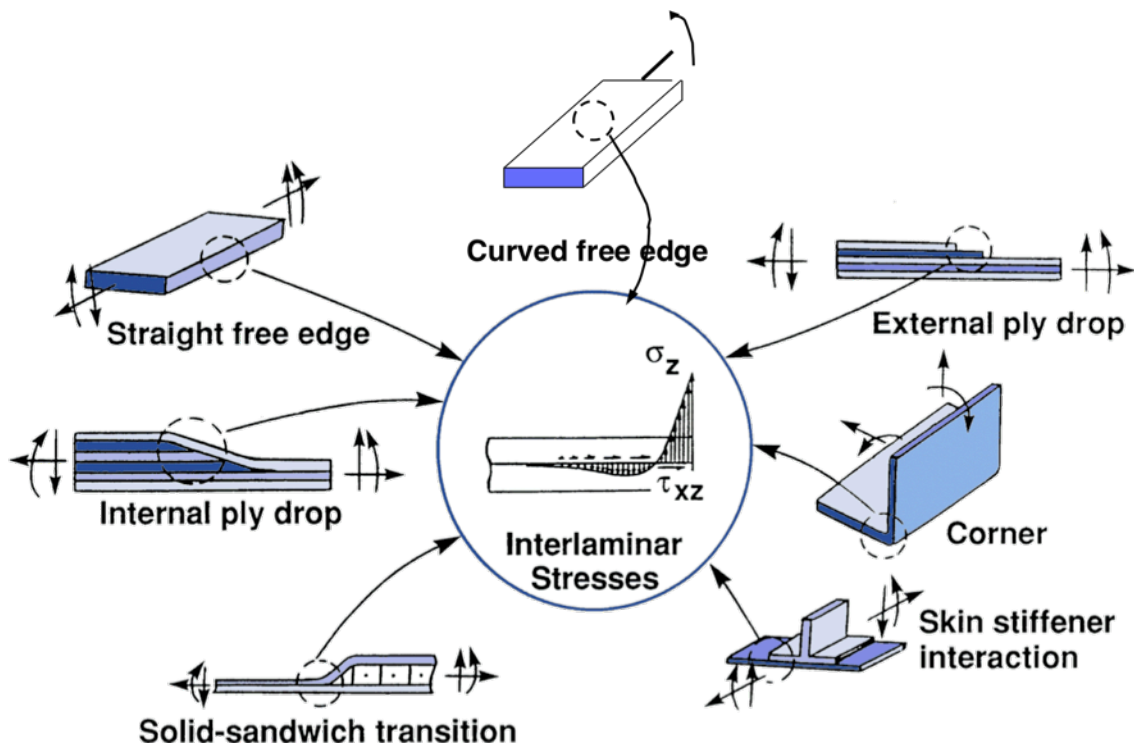


Figure1: Sources of delaminations at geometric and material discontinuities

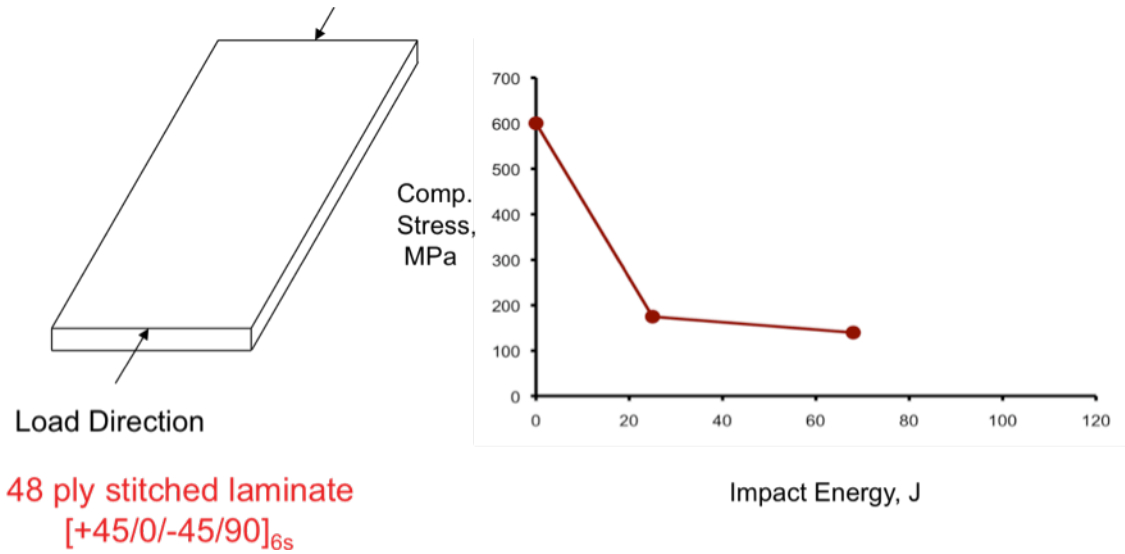


Figure 2: Compression after impact strength for AS4/3501-6 composite laminate (NASA Langley Test data, Ref. 1).

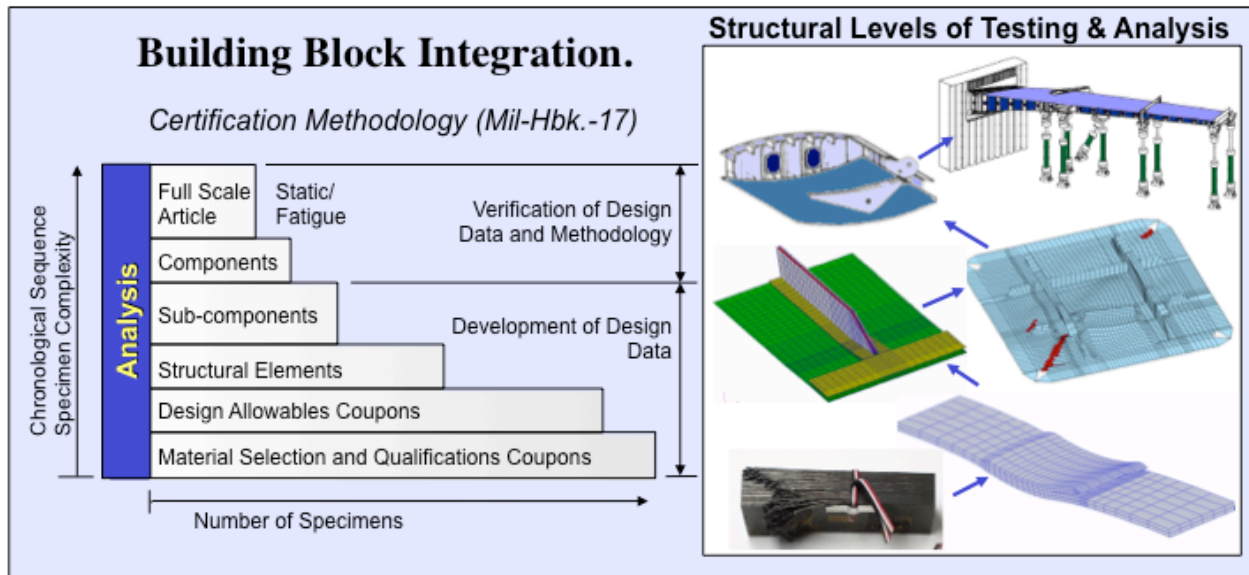


Figure 3: Building block test approach

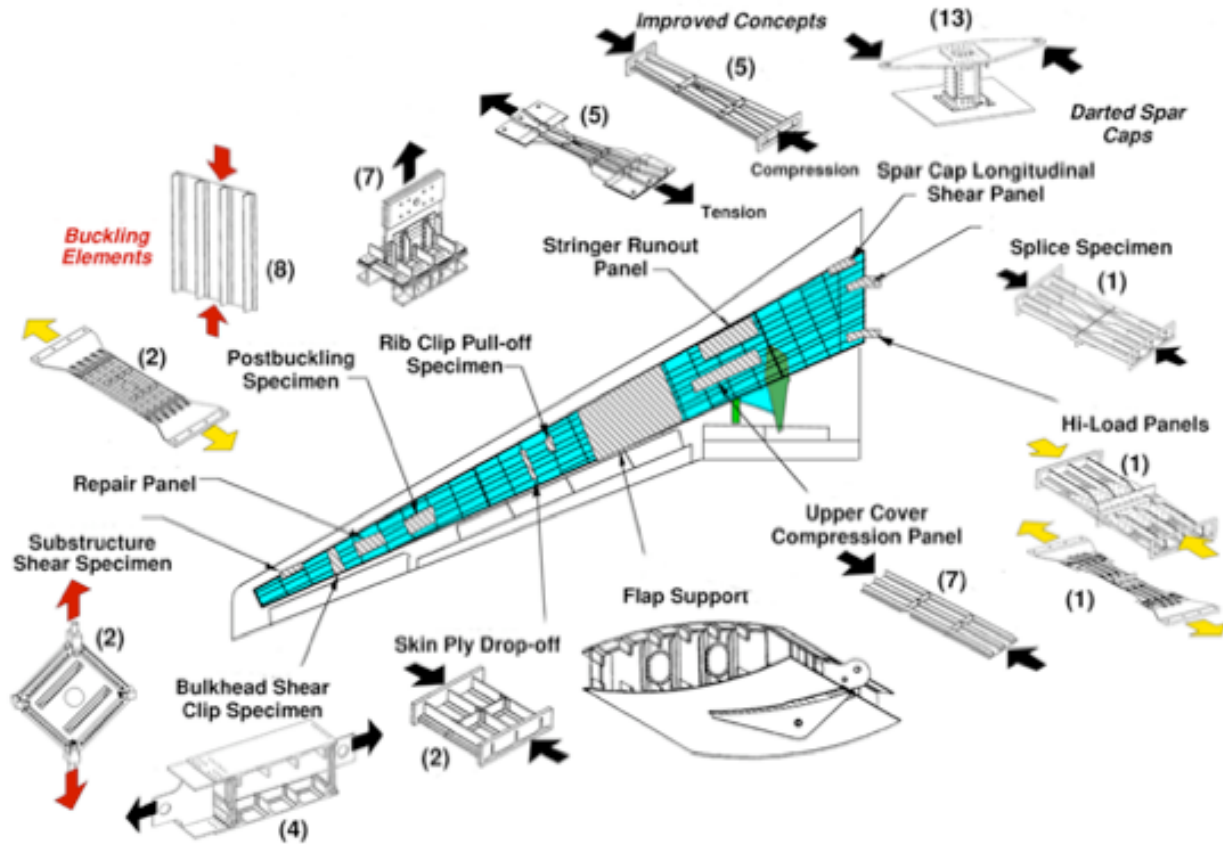


Figure 4: Design development test specimens (numbers in parenthesis refer to the number of replicates for each different specimen) - (See Ref. 1)

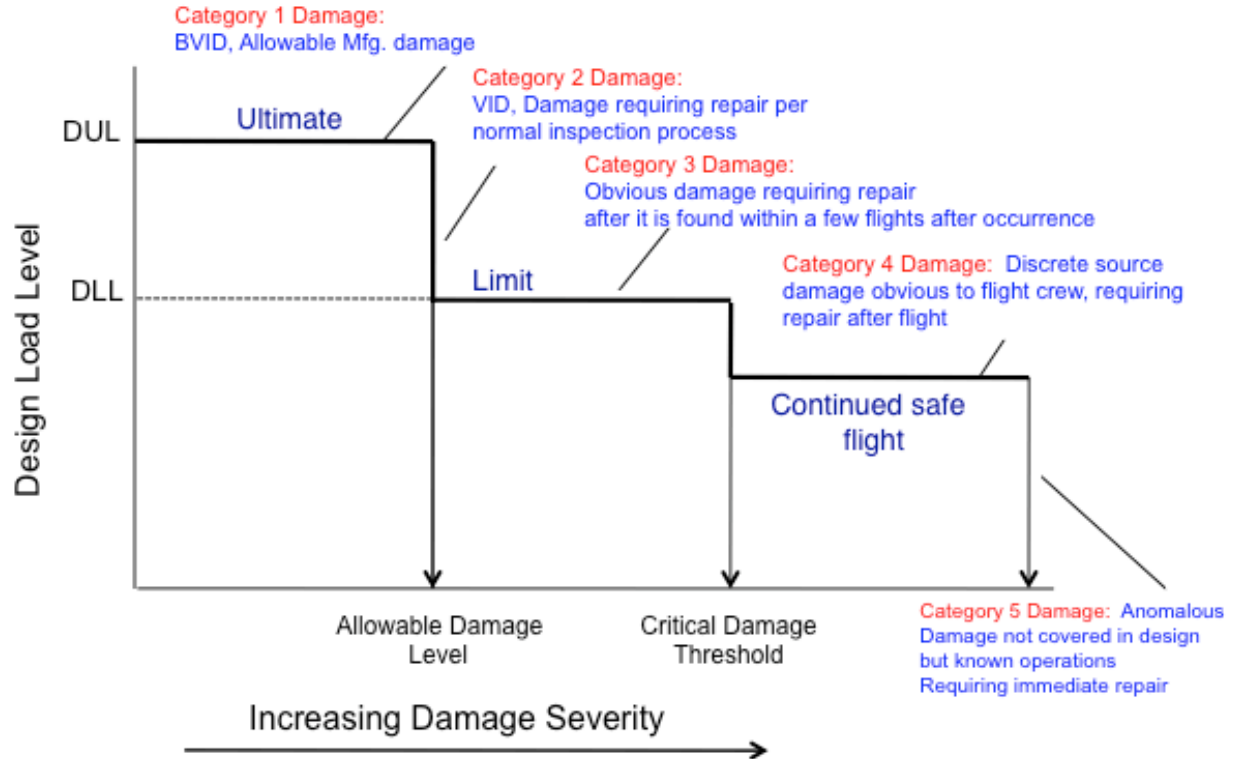


Figure 5: Damage tolerance philosophy currently used by airplanes. ($DUL = 1.5 * DLL$. The structure has to sustain design ultimate loads in the presence of non-detectable damage) (ADL – allowable damage level, CDT – critical damage threshold, DUL – design ultimate load, DLL – design limit load).